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"RESEARCH ASSOCIATED WITH THE LANGLEY
8-FOOT TUNNELS BRANCH"

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By

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The following recording is that of a lecture entitled "Research Associated with the Langley 8 Foot Tunnels Branch" by Dr. Richard T. Whitcomb, National Aeronautics and Space Administration, Langley Research Center, Hampton, Virginia. The lecture was presented at the Ames Research Center, October 21, 1970.

To give you some feeling for what is confidential and what isn't, most of the figures that are confidential are marked so and the ones that aren't are not stamped. There's one problem-figure, that when I get to it I'll indicate it is confidential, we forgot to stamp it.

The talk will be broken down into a number of parts; I expect I'll talk about two hours. The first half of the talk will be primarily involved with the supercritical airfoil and the second half with some of the other work we're doing. If some people are only interested in the airfoil, you can get a feel for it when I'll be talking about that. The layout is on the first slide. The supercritical airfoil experimental theory-then the applications of the airfoil. Of course, as soon as you put it on the wing, it becomes a supercritical wing, and we are now involved with three flight demonstrations on an F-8, T-2C, and the F-111. I'd like to go into that a bit. Then more recently, in the last month or so, we've started on an advanced technology transport. We've been working with refinement of the area rule; I'd like to go into that somewhat. And also we've been doing a substantial amount of work on engine installations. Airplanes have to have engines on them, and the installation becomes a considerably more difficult job than at the speeds we're working with now. Then, we've been working on improving the stability of swept-wings and a substantial amount of work on trying to improve the ability to get wind tunnel data over the Mach number regime; so this is an outline. Now all the work I'll be describing is not, has not, been done at the 8-foot tunnels branch, and where it has not been I'll give credit to the outside organizations.

The next slide, please. The first discussion will be on the supercritical airfoil, and here's a very brief (have we cut something off the bottom there? By golly, that viewgraph didn't reproduce.) I should have checked it, but anyway, I've only got half a supercritical airfoil down at the bottom. Up at the top, we have the flow about a conventional airfoil. At Mach numbers approaching 1, the flow above the upper surface becomes supersonic and terminates in a shock wave. The shock wave itself is not the principle culprit, but the pressure rise through the wave causes separation of the boundary layer which, of course, causes increases in drag, unsteady lift, and usually controllability problems. The pressure distribution associated with that supersonic or mixed flow phenomena is on the right; typical acceleration of the flow to the wave then deceleration behind the wave. Now, on the supercritical airfoil, we substantially reduce the curvature of the upper surface of the wing. You can see at least that part of it, and this reduces the extent of the shock wave and also weakens that wave. In addition, the special shading behind the wing, which you can't see on this picture controls the boundary-layer separation. On the right, the representative pressure distribution for a supercritical airfoil is shown. The pressure distribution is quite flat on the upper surface, and then also you can see, because of the camber we have on the rearward portion, a substantial rearward loading on the airfoil. Some people have called this an aft-loaded airfoil. Now I want to emphasize that the flat distribution is for the supercritical case and not for the subcritical case. This is not the flat top type airfoil that people have talked about at times. At subcritical speeds there's a very large peak near the leading edge which I'll get into in substantial detail latter. This is just a quick run-through as to the general approach to the supercritical airfoil.

Can I have the next slide, please. Here is a historic development of our airfoil. We started out with the flat top, the rearward camber, and with a substantial curvature on the lower surface, and at this point in time we felt that the only way we could control the boundary layer separation through the pressure rise of the shock wave and the final recovery was to put a slot in the airfoil, and this put high energy air under the upper surface boundary layer to energize it in the region behind the slot. Substantial work was done on this airfoil including through the magical testing, but when industry tried to put it in an airplane, they found that they could not hold the close tolerances of the slot. I'd like to emphasize that this is all transonic flow we're talking about, and that slot had to be very, very precisely maintained. I'd like to compare that slot to a curved transonic wind tunnel with a variable total head across the throat, and if the slot were not just right the flow would not go into the slot, jump right over it, and we have a worse airfoil than without the slot. So we decided we'd have to develop something that did not have this very critical item, and so we developed in 1966 an unsloted supercritical wing. Now, the unslotted wing is similar in its nature to the slotted wing in the sense that it's flat on top reduced curvature on top with aft-camber; however, this does not provide quite as much delay in drag rise but we're willing to accept the back off to get rid of that slot. I'll get into a discussion in much more detail later on the unslotted airfoil. Now, down at the bottom is the airflow we've been using since 1968. The industry felt the trailing gage was far too thin for practical construction, and so we thickened up the trailing edge merely by rotating the lower surface downward. The basic curvature distribution is the same. This actually helped a little bit on the aerodynamics and substantially improved the structure.

Can I have the next slide, please? This is the way we get out wind tunnel tests of 2-dimension airflow characteristics. I think any of the people here who have worked with wind tunnel programs know how we do that. The model spans the tunnel so that we have no end effects and the flow is as close to two-dimensional as we can get it. To keep away from the wind tunnel boundary problems, we get all of our data with pressure distributions at the center line, and the drag is obtained by a rake which you can see behind the wing.

Next slide, please. Here'a a variation of drag coefficient with Mach number for a lift coefficient for a normal force coefficient of 0.6. The original for a base line, we have a 64_1-212 airfoil shown by the long short dash here. The slotted airfoil which was 13 1/2 percent thick is this curve here. Even though thicker than the original, we had a delay of about 1/10 in the drag rise, a little over 1/10. Now, you'll notice the level is higher than for the original airfoil. This would be expected because we have added skin friction on the second piece of this airfoil. Now, with the unslotted airfoil, the level is about the same, and then we get a drag rise out to this point. Now, you'd say these are just as good, but keep in mind that the unslotted one is only 11% thick whereas the slotted was 13 1/2, so that the delay in drag rise for a given thickness ratio is less. You'll notice that both airfoils have a slight rise in the drag before it levels off here. That's the development of a weak shock wave on the system which persists as long as we have supercritical flow on the airfoil. Now, many of you that have probably been involved in the theory have heard that the theoretical people are trying to design a shockless airfoil, and they now feel that they might be able to do it. Back in 1964, working with the slotted supercritical airfoil, we did develop a shockless condition. That's shown by this dip. The wake surveys, the schlierens and the pressure distributions all indicate that at that point the flow was shockless. Now, you can see what the problem is though. It occurs at one point. Below that the drag is up because there is a shock at these Mach numbers, and so a shockless flow is a very desirable scientific objective, but if it's only for one point it doesn't have much practical application. In all of our further work, we have tried to get the drag down over a wide range rather than try to get it down at just one point.

May I have the next slide, please? I've mentioned that these airfoils are aft-loaded, and of course the immediate question that every aeronautical engineer asks is, "All right, how about the pitching movement which has to be trimmed." And here's a picture of that situation.

The original 64_1-212 has a pitching moment here. This is again for a normal force coefficient of 0.6; pitching moments against Mach number. Here's the level of the pitching moments for the original airfoil, the slotted airfoil, and the unslotted here. The pitching moments are more negative, but here we see another advantage of the unslotted airfoil. We're not trying quite as hard for the unslotted one to get the highest possible drag rise delay, therefore we didn't have to load up the trailing edges much, and therefore we don't have as much negative pitching moment. I'll get into how we solve this negative pitching moment in considerably more detail later.

Next slide, please. Here's a summary of the drag rise delay for the supercritical airfoil. This is the Mach number for separation onset versus lift coefficient. This is the typical buffet boundary curve or yaw (it's called almost anything), but I think you're all familiar with this type of comparison. The original airfoil here is the 64_1-212, and here is the supercritical airfoil. Two-dimensional critical airfoil here. The lift coefficients we need for cruise in this range, that I've mentioned before, is about 1/10. But notice what happens when we go to the higher lifts. The delay becomes greater and the knee of the curve is pushed not only outward but substantially upward, so that you can see here this is 1.3 compared to 1 here. This is particularly important for maneuverability as I'll get into later.

Next slide, please. Well, up to now I've been giving you a rush treatment of where we stand. I'd like to get into the details. This is a schematic of the field above the airfoil at Mach numbers beyond the critical Mach number. It's purely schematic, and I won't guarantee the exact shape of the sonic line. I'll show you later on some theoretical predictions of the sonic line. As you all are aware, in supercritical flow a supersonic field is immersed in a subsonic field, and the division between the two fields is called the sonic line. If we had purely a supersonic flow, we'd have a continual expansion or acceleration of the flow from leading edge to trailing edge. But when we have a mixed flow, the expansion waves that emanate from the leading edge (as shown here by the dash line) come and meet the sonic line. At this point the disturbance reflects with an opposite sign, that is, it becomes a compression at this point and reflects back along here to this point. At this point, decelerates the flow at the surface; it then is reflected off the surface again now with an equal sign because it is a solid boundary and comes to this point again reflecting as an opposite sign as an expansion wave at this point, so that here is a key to obtaining rational or reasonably good transonic characteristics for airfoils. That there are these disturbances that come back down through the supersonic region to decelerate the flow, and we can get a flat distribution such as I have shown in my first introductory chart, even though there is a continual curvature through here. Now that's the basic phenomena, but that's just a starting point. What is the shape that you need here to get an airfoil utilizing these recompressions that does give you a reasonably small shock wave at this point? Well, I think everybody agrees that there are two factors. First of all, you want to cause as strong as possible expansion at this point to start a lot of expansion waves going out this way that can come back here to slow down the flow in this region. And the second thing is that you want to keep a fairly small curvature in this region so that you don't have a very large amount of accelerations emanating from this point that have to be overcome by these recompressions. It's a very, very sensitive balance between what you produce here and what you don't produce here. Now, that's half the story. The other half of the story, it gets much more complex from the theoretical standpoint but is part of the story, and that is that you want to keep the flow just behind the wing moving at very close to the speed of sound, probably slightly supersonic. This keeps the compressions of this region from moving forward to strengthen this wave. If you didn't have this flow moving near the speed of sound, the compressions would move forward, drive the wave forward, and be substantially stronger and more extensive. Now, while this region is keeping away rearward and reduced, some disturbances are coming around that near sonic region and depressing the velocities in this region. This part of the whole phenomena is not as clearly understood as this. This can be worked out with a method of characteristics, and several people have done that. This area here is purely intuitive at the present time. Now Busemann, when I described this idea, just really didn't accept it because he couldn't accept the fact that to be a region here where the disturbances can't move forward, but over here there is a region where they can move forward.

Next viewgraph, please. That is a brief discussion of the field. Now, let's get into more detail on the effect or the influence of the boundary layers. I'll be talking about this continually through the day (through the next two hours) because the potential flow or inviscid solution for the field is not enough. This is an actual measured data that I took the introductory slide from and you can see that the pressure distribution on the upper surface is quite flat over a substantial portion of the upper surface. Incidentially, this is for a Mach number of 0.8 and a lift coefficient of about 0.6. Now, you can see that there is a wave in the system at about the 3/4 cord point, right here. As I've shown in my previous slide, wave is not extensive. Up here you see the region where we maintained merely sonic flow before we get the final recovery at this point. Now, I want to talk about this not only in terms of what we can do to reduce the waves, but what we can do to help the boundary layer. The boundary layer has to go through ad adverse gradient from this point to that point. If you don't do something about it, the boundary-layer theory says the flow will separate. Well, what we have here is that it goes through this gradient and may, or may not, produce the usual bubble, but I'll get into the whole bubble discussion a little later. But then we have a plateau where the H factor for boundary layer can recover. That is there is a mix in this region so that the boundary layer profile gets back to its more steady state, more stable state and then we go through the final recovery which, of course, is absolutely required in this transonic flow, and so the boundary layer does make it through this gradient, but there's one more piece to this thing that I've got to put in and that is in a conventional airfoil, that is 6 series airfoil, that point, right there, is down in here somewhere. It has to recover to a substantial positive pressure, but in the airfoil design that we've been working with, we kept the trailing edge angle of the lower surface equal to the trailing edge angle of the upper surface so that the stagnation pressure at that point required is substantially less positive. It's up in here. Now, that means that the boundary layer has to go from here to the trailing edge. The recovery goes through substantially reduced, and the possibility of separation is also substantially reduced. These are the type of tricks that we've used to keep the boundary layer attached even though we don't have a slot and even though there is a shock wave in the system. Now, that's the control of the boundary layer on the upper surface. The lower surface must also be considered. Keep in mind that we're driving the Mach number up. As you can see, here's the sonic line and all this flow is supersonic here. The lower surface is a much more sensitive problem. The boundary layer on the lower surface has to go through a pressure recovery from this point to this point, and I now get into boundary-layer theory a little bit and explain that in the theory it's much more devastating to go from zero pressure to 5/10 pressure than it is from minus 5/10 to zero, and the boundary layer in trying to get into these positive pressures becomes very, very close to separation. If the flow in this region goes supercritical, we'll get an additional rise in here somewhere, and the boundary layer just plain won't be able to go from that pressure to that pressure. It will separate, and we have experimental data to prove it. However, if we design the lower surface so that it never goes supercritical for our design point, then the rise is only from here to here. Now, the second point I'd like to make is that that shape of that rise is quite important. If you design a pressure recovery that goes straight down like that, it will separate, according to the boundary layer theory. However, we take our rise abruptly and then let it level off. We do this according to the Stratford criteria. This was something that the British at Stratford worked out for getting more efficient diffusers. Now, when we do this, and we do it just the way Stratford proposed, you take your recovery and then keep the H factor very close to the separation point through here, and according to Stratford, you get zero shear at this point and therefore no skin friction. We haven't been able to actually measure any reduction in skin friction for this thing, for this area, but at least theoretically there is no skin friction for that area.

Could I have the next slide, please? This is the pressure distribution for the subcritical case, for the same airfoil that I've been talking about. As I've mentioned earlier, the pressure distribution has a peak near the lead edge. Now in designing an airfoil for the optimum supercritical operation, you've got to keep that peak from going up to such a severe value that the flow separates at subcritical values, and I want to get into that discussion where we designed some airfoils that did just that. Then to get a nice smooth recovery through here and then flat in this region and then the recovery here. Prior to some of the recent developments on supercritical theory, we used a method of analogy. We designed reasonably good subcritical distributions which our experiments told us were good supercritical airfoils and usually, using this method of analogy, we headed in the right direction. Notice, as for the supercritical case, the subcritical case has a quite rapid recovery at the trailing edge. We've experimented with even more rapid recoveries, and we end up with trailing edge separation. This is just about as far as you can push it. The subcritical lower surface is the same as for the supercritical case.

Next viewgraph, please. I've mentioned earlier that in the supercritical cases between the onset of supersonic flow and the design point, we have an increase of the level of the drag, and you must worry about this case even though it isn't the design point. Here's an intermediate point. This is a Mach number of 0.78. Remember the design point was 0.80. You can see that the pressure distribution on the front portion of the airfoil is similar to the design point, but in this case all the positive disturbances converge at an intermediate point, and this wave is just as strong and causes just as much loss as the one at 8/10 Mach number. This in itself is a problem, but here's the real severe problem. Notice that the flow goes back up and goes supersonic again. If you design the rear portion of the airfoil without just the right curvature, this flow will go up, and the Mach number will be just as high here as it is here, and we'll get a second shock wave which then completely stalls out the region here because it doesn't have the plateau behind to stablilize the boundary layer. So it's quite sensitive to how much curvature you put in this region. From a purely, just trying to aim at the highest possible drag-rise Mach number, you want to put a lot of curvature in here, but you've always got to worry about this intermediate point.

On to the next slide, please. I mentioned just briefly the onset of a separation bubble under the shock wave. Now, I'd like to get into that in a little more detail. It's a well-established fact that, as we go through the speed of sound or through the transonic region, the wave on the upper surface does cause a local separation bubble. This is well-defined in Pearcey's work. And this bubble reattaches and so that the drag rise, or the final separation, doesn't occur right at the critical speed. Now, one of the factors that we'd like to work with is to delay the breakdown of that bubble, and in fact in the supercritical airfoil we're doing that. This plateau region, which is important for reducing the strength of the wave, reducing onset from separation, also is important for delaying the breakdown of the bubble. Here is the extreme case. This is the knee of that curve that I showed some few slides ago. This is for a lift coefficient of 1.3 at a Mach number of 0.73, and here's the Mach number at that point, about 1.5, and the rise through from there to there according to boundary-layer theory should have long ago completely and absolutely separated, but it hasn't because there is a large bubble at this point. You can see the effect from the pressure distribution here and in our oil flow studies. There's a very substantial piling up of the oil in this region, but back here the flow is moving rearward again. And I think it's because of this plateau again. Here in a normal case, the boundary layer would go through this region and through a continually increasing recovery, but here it has this plateau and this reduction in the adverse gradient helps to keep the bubble attached.

Next slide, please. I have mentioned that the ideal of airfoil for supercritical applications should have a very severe increase in velocities near the leading edge and then a very flat top. Now one way to get that is to increase the leading edge radius. So we have systematically investigated various leading edge radiuses, and here are some of the airfoils we tested. The airfoil I have discussed up to now, the data, is airfoil 3. I'd like to emphasize that that is the one that is being used in all our applications. We've tried an airfoil with less leading edge radius and two with more leading edge radius. We really shot the works on this one. But you can see, we can flatten off the upper surface and put a substantially greater turn in this region. Now, that's fine for the supercritical case, but let me show you what happens to the subcritical case.

May I have the next slide, please? Here's airfoil 3, the lowest one, right through there, that's the one we're using. Airfoil 1 bombed; airfoil 2 wasn't too bad, some increase in the level, gradually decreasing down into this region. Airfoil 4 which has a smaller leading edge radius you see has a higher level here, but most of all it has a higher level right in here because we're getting some shock losses right there. So that we think we've arrived at about the optimum leading edge radius. More recently, we tried a leading edge radius slightly bigger than number 3. Just to prove we were on the optimum, and it had characteristics very similar to 4, so I think we've gotten pretty close to a compromise leading edge radius.

Next slide, please. Here's the pressure distribution for Airfoil 2. I've got two angles of attack on there because neither one of those is right near the design point. Follow the circles through. Notice we peak up near the leading edge and then gradually recover before we get to the wing which is just what the theory says we should do to get the highest possible drag rise Mach number or the lowest possible wing drag with that design point.

Now, the next slide, please. But notice what happens to the subcritical distribution for that case. Very peaky near the leading edge. Excessively peaky, and that's why the drag level is up at subcritical speeds.

The next slide, please. Up to now I've been talking about airfoils that we've designed to delay the onset of the drag rise or separation. There's another way to use this whole supercritical development and that is to thicken the airfoil to allow the structural weight to be reduced or, to get more volume, to increase the aspect ratio, all the things the thick wings are needed for. In this case it wouldn't aim at a higher Mach number though. This is an airfoil that was designed by Palmer of North American, and it's 17% thick and has the same drag rise as a base-line 12% thick airfoil. I'll get into a discussion of that in more detail a little latter.

Put on the next one, I'm going to get a glass of water. So far I've been talking about cambered airfoils, but the same phenomena or the same technology can be used to develop better symmetrical airfoils, and these can be used for helicopter blades, for tail surfaces, anywhere where a symmetrical blade section is needed. Here's one we've investigated. As you notice, as with the other cambered airfoils, we have a reduced curvature in the mid-region compared to conventional with the increased leading edge radius and the curvature here. But in this case, we don't have the upper surface and lower surface parallel, as you can see.

On to the next slide, please. Interesting to compare the thickness distribution for this airfoil with some of the other airfoils. Almost universally the helicopter designers use 0012 or 4 digit series airfoils. There is shown here a 4 digit distribution of thickness with length. The best airfoil for delaying drag rise that we've ever developed before was the 16 series airfoil which was used in propeller blade sections. And so here I've compared the (and also the supercritical airfoil here too). Interesting to note the 4 digit series has its maximum thickness forward here and then covering here, and then the 16 has its maximum thickness well rearward. Now the supercritical, you see, has something approaching a 16 here and something approaching the 4 digit here, but the big difference is the substantial reduction of curvature in this region compared to either one of them.

Could I have the next slide, please? Here is the variation of drag with Mach number for $c_n = 0$, and the 0012 section has a drag rise at about 7/10 or there abouts, while the supercritical one has one at about 0.82, just for zero lift however.

Next slide, please. Now we run into a problem though. The helicopter people haven't been using 4 digit series airfoils for the last 25 years for nothing. Here is the delay of a tenth or more at zero lift, but notice the supercritical airfoil onset of the drag rise or separation breaks off at this point whereas the 0012 keeps going. The 0012 is used primarily because it has some of the best maximum lift characteristics we've ever seen in airfoils, that is without flaps. And our objective, at the present time, is to get at least some of this delay but to get the maximum lift back up like it is on the 0012. This work is now under way. As with you people, we have a number of people working with the Army (at Langley) and two of those people have been assigned to developing a good helicopter blade section.

Next slide, please. This is some of the first theoretical development that we've done on the supercritical airfoil. This is some work by Yoshihara and Magnus of the General Dynamics Corporation, and it's a comparison of their theoretical calculations with the experimental results that I've already described. Now I want to go through this in a little detail because he's got a number of curves on here. This is for Mach number 0.8 which I've already used myself. Now, the theoretical curve is here, and he's shown the data for three angles of attack because we didn't happen to have a curve that corresponded exactly with his calculation. And the point to be made is (as you can see with-this is a steady progression to this point) and if we had a higher angle and it was very probable that we could match the theoretical computation in this area pretty well in this region. Except for the level, he's getting the same sort of diffusion here. If he han't gotten the right distribution, we'd have been in sad shape cause that's all subcritical flow there. But I think that this curve does show that his method is reasonably correct because we are getting this flat distribution here. But notice what happens at the back end. He does not have a boundary layer in his computations, and so the supersonic flow just keeps expanding around that curvature that we have, that increased curvature, we have near the trailing edge and the a sudden recovery at this point, so that in this region there's no relationship between the theory and the experiment because the boundary layer isn't there. And notice this big difference here. The boundary layer substantially thickens as it goes through this gradient, in fact it's just on the verge of separation here and so it fills in a large amount of the cusp. The theory doesn't predict that and so there's a very large disagreement between the experiment and the theory.

Next slide, please. This is an interesting plot. Yoshihara's theory does predict the field, and it very strongly indicates the very extensive nature of the supersonic field about the supercritical airfoils. In the sonic line, coming around here, (but keep in mind that this is all imaginary back here) that only the front part is really meaningful.

Next slide, please. Recently (I should have mentioned, we funded that work of Yoshihara's and we are now funding) you might just keep the top in it and as I start talking about it, I'll remove the bottom of it. We've now been funding, or we started to fund, Garabedian and Korn, of the New York University Mathematics Department, and they are working to try to achieve a shockless airfoil using the houograph method. And I'd like to point out that, where Yoshihara's method was using the unsteady solution trying to converge on a steady state, this method is a direct solution of the flow equations. Now, this as I mentioned, the Mach number is 0.80 and the lift coefficient is 0.7. Now would you move it up so I can see the entire picture. I think that-there. I set forth for Garabedian and Korn certain definite ground rules; after all they're mathematicians, and I said all right, you can develop a shockless airfoil but it has to have certain characteristics or it's just a curiosity. The airplane designer has to have an airfoil that meets certain criteria. Well, the first one I gave which was a real rough one was that he had to design for a lift coefficient of 7/10, and the Mach number I'd let him take anything he wanted but here it had to be a lift coefficient. Zero lift cases are very interesting theoretically, but not many airplanes fly at zero lift. The second point I made was that he had to have a reasonably round leading edge because we had to have a reasonably good off design condition. The third point I made was that the 60% station have some reasonable thickness. You can design some beautiful airfoils that come in here with no structure. So he set out with those ground rules and came up with this airfoil shape. So far it doesn't have a boundary layer. Whether he'll get a shockless flow when he has a boundary layer is another question. But this flow, according to him, is shockless, and of course that's indicated by the fact that none of these things are all, all these characteristic lines are not piling up at this point. But the relationship between his pressure distributions and the airfoil shape with what we've been arriving at experimentally is quite interesting. I'd like to point out that as soon as he puts the boundary layer in, this is going to curve down some more, and the relationship with the experimental airfoil will become even closer. He's got a problem here; you've probably noticed this. That curve has a cup in it here. Nothing ever existed in real life where the pressure distribution did that. He knows it, and he's working on it.

Next slide, please. (At this point, Vernon Rossow, of Ames, inquired as to what theoretical approach Garabedian used.) You'll have to speak to somebody else about that. I am not a mathematician, I've been assigned the job of monitoring this contract but I have a co-monitor, Dr. Garrick, of Langley, who is probably one of our best mathematicians, and after he looked it over he said "My god, this is the most complicated mathematics I ever saw." So believe me, I don't understand it.

So far, Yoshihara has tried to put the boundary layer in his calculations, and it just doesn't work. It bogs, and he knows what the problem is. The H factors in the shock wave go up to values that indicate separation, and he has no way to predict what happens in the bubble that forms. And then, of course, the other part of the problem is that right near the trailing edge on an aft floated airfoil, the gradients become very steep. The theory predicts the flow will separate, and indeed it does at the last 1% of the airfoil. But it usually makes the theory bog. Now, here's something we forced to make work for a subcritical case. And I just want to show why we have to put the boundary layer in. This is the same pressure distribution I showed earlier, and we predicted using Van Dyke's method, the pressure distribution for the case without the boundary layer and the case with the boundary layer. Now as you can see, that there is a very definite affect, particularly in this region, but more importantly in this region. This is the region that's always bothered me. Because this is where the pressure distributions tend to peak up a second time. But notice when you put the boundary layer in, we don't get that second peak because the displacement thickness of the boundary layer in this region.

Let's see, what is the next slide I have? Is that the one that has a lot of listing. Let's take that off. Would you take that off, please. All right, this is a good break point. Dean Chapman, of Ames, asked the following question. "Thinking about how sensitive you stated were the affects of Mach number and how important it was to have the boundary layer in these calculation, wouldn't it also might be important that the Reynolds number be taken into account? In the sense that if you had an application for which the Reynolds number was very much different than that of you experiments, you might have to redesign the airfoil a little bit to make it right for the higher Reynolds number."

Yes, I agree with you 100%, and I have a whole discussion on the affects of Reynolds number in that topic called wind tunnel testing problems. But to give you an answer for the time being, both the theory and the experiment indicates that for this supercritical airfoil the lower surface separates at the usual wind tunnel Reynolds numbers to get the lower surface to work right. Now, some people may say that's dangerous; but to keep in mind that in research we're not designing good airfoils for wind tunnels. We're supposed to be designing good airfoils for flight. Therefore, we have forced the shape of this airfoil into a shape that is a bad wind tunnel airfoil in order to get a better flight airfoil. There is a very powerful affect to Reynolds number, to answer your question.

Why don't we take a break for about five minutes?

(A question concerning wind tunnel wall corrections was asked by someone in the audience.)

Yes, at the present time we don't believe the wall corrections. We get very large wall corrections, which when we use them, give a slope to the lift curve that go practically to infinity. So, we don't believe them. Yoshihara in trying to compare his theory with experiment, just used the nominal values of the angle of attack, and they don't agree at all with the angle of attack that he picked. He did the calculations for zero angle. He used data for 1 1/2 degrees, which when corrected goes to minus a half, and so we don't know what the lift correction is at these speeds, but keep in mind that as long as you get your data by a survey rake, you're not too bad off. If you get it by force data, then what is the resolving of the normal force into axial force, and you get answers that can scatter all over the place. But remember the drag is energy loss. If you can measure all the energy loss, you've got the drag.

I forgot something on my discussion on 2-dimensional airfoils, and I should bring it out right now. In all of this work, of course it's all experimental, but we somehow have to quantize what we've done. We don't have a theory, an aerodynamic theory that will quantize it. And so we've done it the way they've done it on the old 4 digit series, by quantizing geometrically. In the back of the printed figures which I've left about 22 copies here on, I have the equations that we used to quantize this shape, and the equations are fairly simple straightforward equations. The front end of the airfoil is a parabola with the parabola tangent to the leading edge at the 45% point, with the slope and curvature equal at that point, and the rear portion is a continually increasing curvature defined by the equations. Now, those equations have now been used on defining wings for at least two applications of the airfoil, and in both cases the wind tunnel results indicate very satisfactory characteristics, so the equations while not elligent do do the job. Let's get into the applications. We are working on a number of applications of the airfoil, and of course as soon as you put them on an airfoil they become wings as I mentioned earlier. In commercial use, probably its most important application is going to be for a long range transport where we can increase the speed, and speed is important as I'll get into in a little while. For intermediate range transports, speed is probably not important; probably weight reduction is more important and there we could use a thick supercritical airfoil that I've showed you. For business jets and helicopters, of course, speed is important. We've got to get the helicopter speeds up. You might ask, helicopters are not transonic vehicles, but forward or advancing tips of a helicopter rotor does get into transonic problems, and if you can use a blade on that that has a higher speed, then the whole aircraft can fly faster. For military applications, it appears that its most immediate use will be on variable sweep fighters. And here we've run at least five months of wind tunnel testing on applying this to an F-111, and in the next month we're going to start applying it to the F-14. For bombers, search and attack airplanes, weight reduction is probably where it will pay off, and two of the bidders for the B-1 contract tried supercritical airfoils and they did substantially improve the cruise, the range factor. They ran into a problem, however. The airplane has to be teamed with a KC-135 and therefore can't fly at the higher speeds that this airfoil lets them go to. So it was not used on the B-1 design. I think you're all aware that the United States government flys a number of high altitude reconnaissance aircraft. The airfoil can be used on these aircraft to get a substantial increase in altitude and we've run wind tunnel tests on one reconnaissance aircraft which were very, very, highly satisfactory.

Next slide, please. Let's turn to the application for a higher speed long range transport. This is a picture of a wind tunnel model of a flight test vehicle that will fly early next year, probably in March. We're using the F-8 airplane as a test bed. I'd like to emphasize that this wing will never be retrofitted to the F-8. The wing is designed for a high-speed transport. The wing has the aspect ratio, the thickness ratios, and the various other characteristics that are required in a transport wing. The airfoil shapes, while you can't see them in this picture, are out in this region the same as the two dimensional airfoils I've been describing earlier. Notice, however, we put a glove out in front here. This is required to solve the very severe three-dimensional problems that occur near Mach one for a lifting wing. Now the first thing it does is to smooth out the area distribution for the whole airplane, bu t that's not the primary thing. We've got to get something, a wing in which all of the lift doesn't pile up in this region. It's good to have R.T. Jones back in here, because the first time I ever ran into this was in looking at his theory for 3-dimensional wings, and he showed that as you approached the speed of the sound, the lift all ends up right here. So it's still a problem with us, and the way we get rid of that is to completely unload this. This upper surface here is flat, there is no camber in it, and we tried, by a very substantial amount of incidence in this region, to get the lift up here. And again I'd like to go back and use some of R.T. Jones's theory. Jones proposed some years ago that you get the minimum wave drag at supersonic speeds if you could distribute the lift so that it was elliptic in shape from here to here, as well as from here to here. Now I know the Jones theory doesn't apply to transonic operations, but since transonics is half-way between subsonics and supersonics, there must be some validity to the theory. So I'd say I'd get a better wing if I tried to distribute the lift forward up here, and that's why that glove is here, and believe me it helps, as I'll get into in a little while. Now this wing for the airplane is being built by North American at Los Angeles, and it's supposed to be delivered to the NASA-Edwards people by the end of the week. But we've got a lot of work, we've got to proofload the wing, we've got to do a lot of things, instrument it, get it attached to the airplane before it flys, and so don't expect the wind tunnel tests next week. Somebody always asks about this bulge back in here. That is an area rule bulge, but it's not the bulge required to get the optimum area distribution. I'll get into that bulging later. But what we found was that, because the vertical and horizontal tails are right opposite each other at this point, there's a bump in the area diagram with a very sharp corner on it, and at the Mach numbers we were testing at, the flow in this region separated because of a very strong wave here. So what we did was to add material in this region to smooth out the corner of the area diagram at this point. Then up in this region, with this new wing, the structural tie points that we have to tie the wing on were sticking out in the air stream because this wing is further forward than the original wing. So we have to fare those over with a fairly hefty fairing here, and it seemed logical to just continue the fairing between that one required here and that one required here. It is not something to get the optimum area distribution. I want to emphasize again.

The next slide, please. Now for the people that are more interested in the more practical problems than theoretical problems, here are some of the objectives. Of course, we want to demonstrate that the wind tunnel is giving us the right results. We also want to establish how this airplane operates way beyond its design points for maneuver and speed margins, say at supersonic speeds and at very high lifts. Same thing with this oft-design performance. We ant to see how the lateral controls work and high lift system works, and finally how sensitive is the surface of a real wing with this supercritical flow, and so we want to get out and just try it out in real life.

Next slide, please. Here's an oil surface flow for that wing at a Mach number of 1 and the angle of attack required for cruise lift. On the inboard region, you can see we've got a fairly smooth flow. There's a little bit of trailing edge separation right in here which we still have to work on, but all in all, that's a flyable wing.

Next slide, please. Now here's the drag of the wing on the F-8, but it isn't the drag of the total F-8 with the wing, because at the Mach numbers we're talking about without the proper area ruling there are very strong extensive shock waves associated with the area development and not the wing. And so we tested an equivalent body of the airplane to get the wave losses and then subtracted the drag for that out of the drag for the airplane to find out how much wing drag we had, that is due to the local flow in the wing, and this is the result. You can see that for near the cruise lift about 4/10, there is a creeping drag rise due to the development of waves as I discussed with the 2-dimensional case, but there's no abrupt increase up to Mach 1. In other words, the flow hasn't broken down yet.

Next slide, please. Here are some of the pressure distributions on that wing, and I've mentioned something about the need for the glove and trying to distribute the lift properly, and here is the pressure distribution on the glove and you can see that we've got a substantial negative pressure well forward. This is if you took the straight line leading edge and extended it here. This is the reference line here. So all of this lift is out on the glove, and then there's a progressive or isentropic recompression and then some abrupt recovery near the trailing edge. Now, as you go out to the cover of the glove, with the main panel, notice that the wing loads up here. If you calculate what the load distribution should be, using a subcritical theory, you find that it will always tend to load up where that highly swept glove meets the lesser swept panel. Here we can see it with the supercritical case. Now as I go out you begin to see a very interesting phenomena going. Incidentally, here's that final recovery that's characteristic of the 2-dimensional airfoil.

On to the next viewgraph, please. Now we're going further outboard, and this is the next station. Now this forward region is extending rearward, and we get the final recovery here. Notice this one beginning to look very much like the pressure distribution for the

2-dimensional case at a Mach number somewhat less than the design Mach number. Then we go on out beyond the mid-semi-span and we get this thing. The two gradients merge. Now the big question, everybody looks at this data and says "Do we really have an isentropic recovery at this point?" I don't know, but it certainly looks like it. So I'll leave it up to your imagination whether it really is.

Next slide, please. Now we've got something that looks very much like the design point for the two-dimensional airfoil, and then finally at the tip, notice how this goes down to a zero velocity at this point, and this is due to the fact that the wave off the corner of the tip is crossing at this point, and you get two compressions converging here, one off the main flow of the wing and another off the tip, and the pressure is driven down to zero velocity at this point, although the flow doesn't really separate at this point, that's quite a steep grading there. I think there's some bubbles phenomena associated with this. It's the only way that thing can hang on.

Next slide, please. This is the span load distribution and all that's put on there is to show it's about elliptical.

Next slide, please. I've mentioned that there is a second way to use the technology, and that's to go to a thicker wing but at the same Mach number, and the Navy is particularly interested in this approach because it will allow them to get more range out of their search airplanes and such airplanes as that. So in cooperation with NASA (NASA is providing the major portion of the funding) we are going to test a thick supercritical wing on this Navy trainer. Now this is obviously not a high performance airplane, but it's good enough for testing this thick superciritical wing. The drag rise Mach number for the original airfoil, and for the thicker airfoil, is about 0.73. This wing is being built by the Columbus Division of North American. It also should be completed this week, but they expect to fly next month mainly because they don't have the structural problem we have on the F-8. They're just gluing balsa wood on the original structure, so it's just a matter of taking it up and flying it.

Next slide, please. A third application that we hope will be flight tested. The money is in the budget of the Air Force, but they haven't given the money to General Dynamics to do anything with it yet. This is the application of a supercritical wing to the F-111, to demonstrate its capability of improving maneuverability. I'd like to point out that the program we're talking about is quite parallel to the F-8 program and the T-2C. This is not a pre-production prototype. This is a proof of concept flight test. It's a technology program rather than the usual type of program that goes into new airplanes. This is just a picture of the F-111. If you haven't seen a picture of an F-111, you haven't read a newspaper in a number of years.

Next slide, please. The first application we worked on for the airfoil was the F-111, and about 2½ years ago we went through a fairly extensive program. We tested not only the original airfoil but also a more cambered airfoil because to improve maneuverability just a plain conventional camber might work. Then this is the supercritical airfoil we tried, and then we tried various flap deflections of 5°, 10°, and minus 5° to get a complete picture of, in this case it would amount to, an effective change in the trailing edge camber.

Could I have the next slide, please? Here's some of the results we got in that first phase. This is for 26°, which is the angle they use for cruise, and lift coefficient for cruise. Here's the cruise Mach number for the basic F-111, and here's the cruise Mach number with the supercritical, getting about the same delay as we got with the two-dimensional airfoils. You'll notice that the 4/10 camber didn't help much on cruise Mach number, and that when we deflected the flap 5° we got a little worse, so we feel that the camber we had in was pretty good for cruise.

Next slide, please. Now, I mentioned that while it was very helpful for cruise, the big gain we though would come in maneuverability, and here is the same sweep angle for a lift coefficient of 9/10, again with the same four sets of data. The original wing here, the 4/10 camber did help here. But now the supercritical airfoil is substantially better using 4/10 camber and with 5° flap deflection, it helped open the zero degree deflection. So this is the sort of gain you can get in the drag for the maneuvering case.

Next slide, please. Here is a comparison of the buffet onset for the baselines and the supercritical. We determine buffet by putting a strain gauge in the roof of the wing and measuring the oscillations of the bending moment at the root, and we get a diagram (this is a little typical diagram we get down here) where this is the weighted fluctuations versus the lift coefficient here, then we pick off a point where the thing breaks up. The original airplane, it's kind of hard to say where it breaks up, whether it's here, or here, or here; but for the 4/10 camber it's quite distinct. So going up to this same sort of plot I had before, the buffet onset plot where it's lift coefficient against Mach number, the original airplane is a solid line, and the circles are what they obtained in flight. For 4/10 camber, you see, it helped. But then for the supercritical, up in here, it is substantially better than just plain increase in camber.

Next slide, please. Now, this slide is not marked confidential, and it most definitely should be marked confidential. So if you get a hold of the printed copy, treat it as confidential. Since those original tests, the Air Force-that was done as a NASA program, just to demonstrate to the military service what we could do with a variable sweep fighter. Since that time the Air Force became quite interested and decided on this program for putting a wing on the F-111, and at this point (since we're going to build a new wing--in the original work is the same thickness ratio, the same plan form. Just to sort out the pure effect of the airfoil shape. But now the Air Force says, "We'll put a new wing on, but there's no point in sticking with the original plan form and thickness ratios; let's build the best possible wing for getting maneuverability that we're using supercritical airfoils as a starting point." And so they have reduced the aspect ratio somewhat and increased the wing area, keeping their root bending moment the same. Their objective here was, you see, we can hold the range--we'll get more range than we need; we'll get more range by delaying the drag rise, but now let's use that, keep the range the same and do as much as we possibly can to get maneuverability. One way to do that is to reduce the aspect ratio. So this wing that I'm going to talk about now has a lower aspect ratio. The original, the baseline here is the F-111D. Now the D is fairly long in their change in airplanes, and it has the new inlet and this data does not compare with the data on the other slides mainly because that inlet has a lot of drag. But that's not our problem here. The big discussion is how we get the inlet drag down and has nothing to do with the supercritical wing. Now, this is the F-111 tip. That's the transonic improvement program. It's now called TAC, Transonic Aircraft Technology Program. The wing sweeps are different here. This is the best sweeps we have for this case, this is the best sweep we have for this case. We took the optimum in both cases. This is for Mach number of 9/10 and for the time being it's untrimmed. Would you raise the slide a little bit. This is the C_D against the lift coefficient, and for the low-altitude maneuverability which they're quite interested in, in this range the drag at a given lift coefficient is cut in half. So the energy maneuverability for this airplane should be substantially improved. Now, looking at the buffet onset, here's the buffet onset for that configuration, original configuration, here's the buffet onset for this one. The airplane is going to be buffet free up to the structural limit of the airplane.

Next slide, please. Now, up to now I have been talking about some of the research vehicles. More recently, the NASA administration decided that we ought to exploit the new wing more fully, and so they asked Langley for a program for a deeper exploitation of the wing, and we proposed a substantial program, and I think a number of you people here are aware of that program. However, it boils down to the fact that we decided to try and develop an advance technology transport which would incorporate the supercritical wing and the advanced area ruling which I'll get into a discussion of a little later. The program, for the time being, is involved with extensive wind tunnel testing and a systems study, or a preliminary design of an airplane by industry. Some day, it might be a test vehicle, but there is no planned funding for the test vehicle yet. Now, why do we want to go to the higher speeds? Why do we need a faster transport? Well, there's always the desire of people to get from one place to another in faster time, but again I'd like to point out that that last hour on a transcontinental flight gets to be a long hour, and if we can cut an hour off that flight, it would be a more pleasant flight. That's the passenger appeal. Economics, if you increase the speed of your airplane and don't change anything else, the cost of flying goes down for two reasons. First, the range equation, the L/D times Mach number is increased, but more importantly the utilization of the aircraft is increased, and both of these terms come into the DOC equations. And finally, there's an interesting fallout here. If you can increase the speed high enough, the airplane of course will fly at a higher altitude and therefore will fly in new corridors and won't have to be piled up in the existing flight corridors.

Next slide, please. This is a picture of a wind tunnel model of the version of this advanced technology transport as we have it as of now. The wing is the same as the flight test on the F-8; the fuselage is coke bottled; I'll get into that in the next slide. As you can see, as yet we have no engines on it, and the tail is merely the F-8 tail. We'll have to put a transport type tail on it.

Next slide, please. Here is shown the area ruling. I want to get into the area ruling in considerably more detail later, but the fuselage is coke bottled, and at least the Boeing people say there is no problem on coke bottle fuselages. They're doing it on the supersonic transport, and they feel it adds very little added weight or added cost, although the Douglas Company has never built a coke bottle fuselage, or even thought about a coke bottle fuselage, and they think it's going to add to the cost.

The next slide, please. Just an idea of how the wing compares with an existing transport wing. To get to the Mach numbers I'm going to--that we're trying to get to, we put a little bit more sweepback in this wing than on the 747. So the gains I'm showing are not pure supercritical wing or pure area ruling. There's a little more sweep to help too.

Next slide, please. So here's the data we got. As you can see, we got a drag rise, a slight drag rise, up to Mach one. Compared to the existing 707's, DC-8's bearing down here, and the air buses here. The people designing the air buses are promising the airlines 0.85 cruise up here on the drag rise, and Boeing 747 because of engine problems has a cruise of about 0.83, but that's not your problem, that's Pratt & Whitney's. Up here we've got some drag rise, as I've pointed out, but all airlines fly their airplanes up on the drag rise a little bit, because speed is more important than L/D in the total operating equations. I've mentioned before that it comes in both on the range factor and the utilization, and checking with the companies they say that they find them up in the drag rise, this drag about 20 counts. That's just about what this is. Another way to look at it, for years we've been using the increment in drag versus Mach number; that is, $\Delta C_D/\Delta M$ of 1/10 for drag rise. That occurs right here at Mach one. So maybe we have an airplane that could cruise at Mach one, if we could get the engine and tail on without too much penalty.

Next slide, please. Now, I mentioned that the new airplane is area ruled on the basis of a refined area rule. As you all know, in area rule the total airplane is related to a body revolution. Take all the cross section areas, bring them down to a body. And then you can calculate what the ideal body should be to get the minimum wave drag. There's a lot of theoretical computations of ideal bodies for supersonic operation. You can do it for minimum drag with a given frontal area of given volume, and yet at transonic speeds there's no theoretical or experimental data on what is the optimum body for near Mach one. I know there is some theoretical work on bodies in revolution, but they don't aim you at the optimum one. So we've done some work in the past year in trying to get the minimum drag body, and this is one of the best ones we've had, and here is that shape of the body up here. Notice it is blunt. We haven't developed the bow wave yet so there's no need to sharpen the nose. The principal thing we want is a low curvature in this region just like for the airfoil, and the area distribution that we're using is here. The front end is defined by this formula. The curvature is an inverse function of the area. For the rear portion, the curvature is a constant. In other words, that's essentially a circular arc area distribution. And down here, we have the drag. At Mach one the drag is about .01 based on frontal area. Since most transports have a frontal area to wing area of about 1/10, that means that the drag at this point (based on wing area) would be about .001 or about 10 pounds of drag. This we can live with.

On to the next slide, please. This is the actual longitudinal variation of cross section area for that transport configuration I showed you a little while ago. Notice it doesn't have the same area distribution that the ideal body does, and this is in there for a very definite purpose. When we tried the area distribution with the ideal body, we had a bow wave standing in front of the wing at about the corner of the glove with the wing. Also the drag rise was substantially more than I showed you on that curve, and this had us stumped until we realized that if we're going to get the optimum at Mach one (at Mach one the curvatures are very sensitive--the drag is very sensitive to curvature particularly in the middle region of the body) that we have to account for the displacement thickness of the boundary layer on the wing and for another factor which

I'd like to go into in a little detail. We have to account for the displacement of the stream tubes on the upper surface. The area rule is essentially a linear theory. On the upper surface of the wing at the lift coefficients that we're dealing with, the Mach number is about 1.3. Going back to those pressure distributions, just convert those pressures to Mach number and you'll get 1.3. Now, at Mach one any deviation of the velocity from the stream velocity will cause an expansion in the stream tube. That's one case where it will happen. Every other case, there will be an expansion of one region and a contraction of another. But in this case, whether the float slows down or speeds up around the airfoil, it will expand. We've got to account for that expansion. Now this is particularly strong on the upper surface. Just going back to one-dimensional tables at 1.3 Mach number, there is a perceptible expansion of the stream tube. Now how much does this amount to? We made some calculations, and the boundary layer displacement is about 5% of the maximum area. Thirty square inch body, you have 1½ inch displacement thickness of the boundary area, and this stream tube displacement is about 1½ square inches. So now if you go back and add 3 square inches to this curve right there, you have a smooth curve just like the ideal body. But we have to account for these second order effects.

Next slide, please. Here is a schlieren photograph for this airplane at a Mach one. That's the Mach one case. Here you see that, even though we've already worked hard with that dip in the curve, we're still getting some of that wave, which in the previous case was a very white, strong, normal wave. Now it has weakened a bit; these two lines are where the wave hit the tunnel wall. Now we've got these two waves here. Let's follow those down here. That's the wave at the final trimming edge point where there's a definite compression at that point which the pressure distribution shows. This point right here is coming off of this little wedge we have here to try and weaken that wave. Incidentally, notice that the tunnel wall is right here. These waves are going to the wall. I'll get into a discussion of that a little later.

Next slide, please. I've mentioned that in order to get up to Mach one we've got to area rule. Right at the present time, the F-8 will fly very shortly has a drag rise that occurs between 0.97 and 0.98, and the schlieren figures indicate this is pure wave problem. So we now have been given funding to add side bearings on the F-8 to produce area distribution very much like the one I just showed for the transport. What we're going to do there is add fill on the sides here and back in here. Now, this is not an ideal area ruling. We'd like to put some of the addition on top of the fuselage, but the pilot couldn't see very much if we did, so we're going to have to put it on the side.

Next slide, please. I mentioned we hadn't put edges on yet, or tails. Next week we expect to get at that, and here is the arrangement we're going to use. It's a copy of the 727 arrangement, two engines of the side, one up here in the vertical tail, a T-tail shown here; the horizontal is a direct copy of the 727. Now, since there's a sting in here which isn't shown, we can't very well have air flow going through this one, so it's just a body of revolution where the tube of air that goes through the middle is subtracted out, and this is the area of the remaining nacelle. These, of course, can be made flow through. The point I want to make here is that the area distribution for that whole rear end of the airplane is identical with the area distribution of the airplane model I showed you a picture of, and so where we have area added for the nacelle we indent the fuselage. We fit the thickness ratios and thickness distribution for the vertical and horizontal tail; so that from this point rearward, it's also a smooth area of distribution. One more point, the airfoil shapes on the vertical and horizontal tail are supercritical, very similar to those symmetrically symmetrical airfoils I showed earlier.

Next slide, please. Let's get into engine installations. I'm going to have to go through this a little fast because we're running out of time. Just to show that we have done some work on aft engine mountings, here is one arrangement. At the lower Mach numbers the drag is increased, but when you go to the higher Mach numbers the drag actually is favorable because the engine fills the hole in the area diagram.

Next slide, please. We've also done considerable work on engines mounted underneath the wing. We've done this on three configurations including the C5 and the DC-10, and we have a little model engine here powered by nitrogen driving a turbine in the fan.

Next slide, please. I'm not going to have the time to go through a very fascinating subject of reducing induced drag, but let me just go through this. This is the drag of adding the engine versus lift coefficient for a Mach number near a cruise, and notice that the drag actually is favorable with no power on. When we turn the power on, the drag is substantially reduced, and this favorable effect we first noted on the C-5 (in our wind tunnel test of the C-5), and there was a very long discussion with the Air Force as to whether it was real. I went back in and did a lot of work with the power case to show that it was still real, so they let the Lockheed Company use this favorable increment in their performance analysis, with considerable trepidation I might say. But when they flew the airplane, the performance that they predicted on the basis of that point actually occurred in flight.

Next slide, please. Now, if power on does help the induced drag when the engine is inboard, it must have a very powerful effect if the engine is out at the tip. So, as a pure research program, we have investigated an engine mounted at the tip of the wing as shown here. Now this wing is uncambered, unswept, so that we don't get all tied in with what is camber effect and what is angle of attack effect, and, of course, we don't have to worry about what the effect of sweep is.

Next slide, please. Here are some of the drag due to lift factors that we get where the original wing is a solid line, the dash line is the ideal, and then we tested the engine at two angles of incidence. The one to look at, the more interesting one is zero where down in here the drag due to lift is reduced by 1/3. If anybody wants to talk about that, I'll be available for the rest of the day; but if I'm going to get this talk in two hours, I can't go into the very interesting details of that.

Next slide, please. For years people have been talking about putting an engine at the rear of the fuselage to have the fan essentially swallow the boundary layer and the fuselage. If you do this you can show that you get a definite improvement in the aerodynamic performance. We're not sure whether to attribute it to an increase in the engine performance or an effective reduction in drag, but because of the energies involved there's an improvement in performance. For years I've thought about getting it into the 8-foot tunnel, but we've never had time, and now the 16-foot tunnel at Langley thinks they might get at it. I offer this particular approach to improving airplane performance as something somebody may want to try here.

Next slide, please. The airplane swept back wing, such as the wing we're going to test on the F-8, has pitch up characteristics like every other swept back wing does. So we've been working on various means of improving these pitch up characteristics. This is the effect of q. Notice that for the pitching moment (this is for 0.99 Mach number) the q of 600 gives you this pitching moment; the q of 925 gives you this. Part of that is due to Reynolds number and part is due to aeroelasticity, and we have substantial data to sort it out, and both of them are helping and improving the situation. What I just want to emphasize there is that the pitch-up characteristics for the airplane will probably not be as bad as the model. That's what everybody in industry has found when they went out and finally flew swept back wings.

Next slide, please. But assuming we do have a problem, how do we fix it? Here's the device that we've tried, and it works. We put a fense or vortex generator on the lower surface. Now this is very similar to the borderline that is on the DC-9, and I've checked with Douglas, and I've applied for a patent on it, so I'm not proposing this as an original idea. I'm merely proposing it as a means for improving the pitch-up of the swept back wing. Here, ours is substantially different from what they have, but the basic phenomena is the same. Here is the original airplane, here is the airplane with the lower surface vortex generator, and the pitch-up characteristics are as different as night and day.

Next slide, please. These are very poor oil flow photographs, but they do show the point I want to make. Without the vortex generator you can see the flow of the oil moving out in this region. Now, with the vortex generator, here it is moving out, and then it stops dead right there.

Next slide, please. I think we'll take another break at this point because I said I'd talk two hours, and I've talked two hours; however, I'm just getting into the discussion on wind tunnel testing techniques and the problems we are working with. It's a subject that's usually involved with specialists on wind tunnel problems, and so I probably ought not to go through it for the whole group but the people that are worried about Reynolds number effects or wall effects or one other effect we're now trying to tie down, humidity effects, these people might want to stay. I suggest we have a break, and after that I'll get into a new session on testing techniques.

So let's get into some of the wind tunnel problems. The question was brought up earlier about Reynolds number effects, and I'd like to show what we're doing to try and simulate full-scale Reynolds number in the wind tunnel. This is essentially the paper that Blackwell gave in Paris a year and half ago, and it kind of summarizes what we've been doing. As you probably have heard, on the C-141 the measured position of the shock wave and the entire stability problem for the airplane in flight was not the same as for the wind tunnel tests, particularly at the higher Mach numbers where a shock wave was present. This kind of shook up the industry, and lots of work has been done on seeing what we can do about this problem. The approach that almost everybody in the industry wants to go to is to build a full scale transonic wind tunnel. We don't have one yet, so what do we do in the meantime? Here is the problem. This is a typical condition for the case where you have a boundary there and then a shock wave. At low Reynolds number, of course, the boundary layer is thicker; and when the boundary layer goes through the wave, the thicker boundary layer becomes still thicker. Now at higher Reynolds numbers, we have a thinner boundary layer, and when the boundary layer goes through the wave its net increase in thickness is less. So what happens? Well, this thick boundary layer pushes the shock wave forward. It's like a wedge pushing on the wave, but here with the thinner boundary layer, it doesn't push as hard and the wave is further rearward. This can cause substantial changes in the pitching moment of the airplane. I'd like to point out as we get into it that other effects occur too.

Next slide, please. In our work we use the airfoil off of this airplane. It's the old P-80, trainer version of it, and we use this as our baseline because it's an unswept wing of fairly high aspect ratio, so we thought the data obtained at this point would be fairly close to 2-dimensional. And we want to do 2-dimensional testing.

Next slide, please. We tested the two dimensional airfoil and got good results at full scale Reynolds number. We could get full scale Reynolds numbers in this case. It's a fairly small airplane. Here's our full scale data, 16.8 million, the transition at 5%, and that solid line is shown here through here. Now over here we have the wake profile at the trailing edge. Now when we went to wind tunnel Reynolds numbers, by pumping the tunnel down (in this case the tunnel was pressurized, and in this case it's depressurized) so it's exactly the same model, just that the pressure is changed, and the same location of transition. Then the shock wave moves forward, as it did on the C-141, and also very importantly notice the velocity pressure recovery and a very substantial increase in drag. Now, our proposal was to make the displacement thickness or perhaps the H factor (we werent't sure at that point, but we went through a systematic program to find out which) the same at the trailing edge, because our oil flow studies indicated that the separation did originate at the trailing edge. Now what we do here is to maintain laminar flow over a substantial portion of the chord, then trip the boundary layer so that the relative displacement thickness at the trailing edge is the same as for the flight article. Now this is relative, remember that; it's the height to chord. When we did that (well, we tried various locations) but we used 40% chord and at the same Reynolds number. Notice now that the experimental pressure distribution and the experimental weight are the same as for the flight Reynolds number. So this gave us a means for trying to simulate the full scale case.

Next slide, please. Here's some theoretical boundary layer characteristics for the same three cases, but it happens to be now for a subcritical Mach number because at this particular--for the boundary layer calculations we don't have a supercritical boundary layer theory. So we had to go back to subcritical cases for the calculations. But it gives you some idea of what's happening to the boundary layer. The original plate data solid line, then wind tunnel Reynolds numbers with the same transition location as this. This is the displacement thickness. And then finally, with the transition moved forward. Now there is a difference down in here, you'll notice, but when we get into the critical area where the boundary layer separates, or will separate (it isn't separated for this case) the displacement thicknesses are very much the same. The same picture applies for the H factor.

Next slide, please. So, we've set up a chart, using again boundary layer theory, and this is what we use for placing transition of all of the test models that we've tested in the 8-foot tunnel. Here we have transition location versus the relative Reynolds number of the wind tunnel to flight. Obviously, this value becomes very low. Oh, pardon me. This value depends on various conditions of the model, conditions of the airplane. Big airplanes, of course this value goes in this direction and the small airplanes in this direction. Now, this curve that we finally worked out is a function of the pressure distribution on the front end, and we get two different curves. If we have this type of pressure distribution, we get this curve; this type, we need this curve, and you'll notice that that curve is very close to what you have if you assumed the flat plate. So, for most of our work for supercritical airfoils where our distributions are like this, we're using this curve. This is a way-off design curve for most airplanes.

Next slide, please. Now, there's one big problem with this method. It doesn't work if you have an adverse pressure gradient in the boundary layer ahead of the transition trip, because then you get natural transition ahead of the trip and usually if the gradient is severe enough you get a laminar separation bubble ahead of the trip. So, it is not a universally applicable method. It works pretty good for supercritical airfoils at their cruise conditions, but it doesn't work at off-design conditions. As I pointed out earlier, we have a peaky distribution for subcritical cases; so we usually test our models with two different locations, one forward for the off-design cases and then one rearward for the design case. But even for the design case we run into a problem. You remember the pressure distribution I showed for the three-dimensional case had a peak up forward here. As we went outward, that peak spread and finally was the same as for two-dimensional airfoils out here. So we have to move the transition in this region forward of the ideal location so that we are ahead of the adverse gradient on the wing. So in this region we're not simulating full scale characteristics even for the design case, although we are out here.

Next slide, please. That very briefly covers the work we've been doing on Reynolds number. If there's any more questions on that whole thing (we've done lots and lots of work, much more than shown here) I can get together with you later. But now let's get into a problem we've just recently run into. We're getting up pretty high in Mach numbers. We've known for years that there is a humidity effect in supersonic wind tunnels, but most people have pretty much ignored the problem in subsonic wind tunnels. But I don't think we should be ignoring it. Here is some data we obtained on the supercritical wing on the F-8 in the 8-foot tunnel at Mach number of 0.98 at a lift coefficient of about 6/10. We have the capability, since this is a closed wind tunnel, of changing the humidity, and we usually test at 120°F and hold it there, and we have the capability of drying. Here's dew point. I think you're all familiar with what dew point is. That's the wet bulb, and we usually after running into this problem some years ago, we continually held our dew point down to 25°. More recently, with this whole subject of how we're going to make a full scale transonic wind tunnel, we decided to get some data at the humidities you might get in an atmospheric wind tunnel, and here's the data. Notice as you go out to a dew point of 60° which for T_o = 120°F is a pretty dry tunnel. The drag is up 60 counts. That's useless data. I mean you can't correct for it. We've gone through the text book, and they've thrown their hands up. They can correct for the effect of humidity on the Mach number in a supersonic tunnel, but they've got one line for the effect of the induced velocities on an airplane or a model of the tunnel. We don't know how to do it. The effect of temperature is over here. We now know the dew point to this 60° case and vary the temperature, and there is an effect as everybody expected. Now that's our disconserted story at the present time, and we've got to accumulate a lot more data, but I just put it up to make you aware of the problem. We've got a handy-dandy rule. Usually this curve starts to go up at the point where you first see flecks of fog in the tunnel. At this point you can't even see the model.

Now, the next slide, please. Going back to the slide I used for the body of revolution, you'll notice that I have three sets of data here. We tested three different size bodies to find out what are the wall affects as we approach Mach one, and you'll notice that all three of those curves fall on each other up to Mach one. But beyond Mach one, they start to diverge. Now, obviously beyond this point there may be other things that happen, but we haven't been interested in anything beyond this point. I just want to emphasize what we've guessed for years, and we have some wind tunnel data to back it up, that up to one we're probably all right but beyond that point (up to the point where the wave off the nose clears the back end of the model) you don't get the right data in a wind tunnel. Well, that's a very brief summary of the talk I plan to give on wind tunnel testing techniques. There' a lot of other work going on, but as I gave in the title this is the work associated with the 8-foot tunnel, and there are committees set up for all these problems, and I didn't want to go into the whole subject.

Are there any questions on this part of the work?

Stuart Treon asked the following: "Dick, relative to wall effects in general, are you looking also seriously at the lifting case for high lift generating shapes?"

Answer: Yes, as part of that we recently tested the F-8 model in the 11-foot tunnel here and had a balance shift, so I don't know what we got out of the test. In addition, we're building a smaller scale F-8 model to test in the 8-foot tunnel, because all I've shown so far is the affect of solid blockage. What is the affect on lift? That's a good question.

Treon then asked, "Is there anyone tackling this from the theoretical point of view as well?"

Answer: Well, when we first started trying to decide how big a model we should use for the F-8 in the 8-foot tunnel, I went to Ray Wright who, of course, was the guy who wrote the original theory on transonic wind tunnels and has been the expert in NASA on wind tunnel wall corrections for years. There are a number of other people throughout the world, of course, and he says my theories don't work at Mach one. So Ray Barger, who used to work for me until Ray retired, is now-I talked to him about it and he's the kind of guy who takes a challenge. He's trying to work out a theory, but he hasn't got one yet.

Thank you very, very much.

EFFECT OF BOUNDARY LAYER ON CALCULATED PRESSURE DISTRIBUTION

$M = 0.6$; $\alpha = -0.1°$